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Advanced Seal Technology Role in Meeting Next Generation Turbine Engine Goals

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Abstract

Cycle studies have shown the benefits of increasing engine pressure ratios and cycle temperatures to decrease engine weight and improve performance in next generation turbine engines. Advanced seals have been identified as critical in meeting engine goals for specific fuel consumption, thrust-to-weight, emissions, durability and operating costs. NASA and the industry are identifying and developing engine and sealing technologies that will result in dramatic improvements and address the goals for engines entering service in the 2005-2007 time frame.

This paper provides an overview of advanced seal technology requirements and highlights the results of a preliminary design effort to implement advanced seals into a regional aircraft turbine engine. This study examines in great detail the benefits of applying advanced seals in the high pressure turbine region of the engine. Low leakage film-riding seals can cut in half the estimated 4% cycle air currently used to purge the high pressure turbine cavities. These savings can be applied in one of several ways. Holding rotor inlet temperature (RIT) constant the engine specific fuel consumption can be reduced 0.9%, or thrust could be increased 2.5%, or mission fuel burn could be reduced 1.3%. Alternatively, RIT could be lowered 20 °F resulting in a 50% increase in turbine blade life reducing overall regional aircraft maintenance and fuel burn direct operating costs by nearly 1%. Thermal, structural, secondary-air systems, safety (seal failure and effect), and emissions analyses have shown the proposed design is feasible.

Introduction

Recognizing the need to reduce costs, NASA has put in place several programs to improve both engine and vehicle performances and lower direct operating costs (DOC). General program goals include:

1. Reduce commercial aircraft direct operating costs including interest by: 3% (large engines) and 5% (regional engines).
2. Reduce engine fuel burn up to 10%.
3. Reduce engine oxides of nitrogen (NO_x) emissions by greater than 50%.
4. Reduce airport noise by 7 dB, or about three-quarters reduction in acoustic energy.

Meeting these aggressive goals for engines to be certified early in the next century (2005-2007) requires significant advancements in each of the primary component technologies—including the supporting mechanical component technologies such as seals. Airlines have become increasingly cost-conscience. As a result, the NASA/industry team is pursuing technologies that show promise of high performance-to-cost benefit ratios. Technologies are being considered that increase engine and vehicle performance, lower acquisition and lifetime costs and reduce engine maintenance.

Advanced engine seals show promise of reducing engine losses and maintaining these performance benefits over the service interval of the engine. New seals coupled with improved analysis codes give the designer better control of engine secondary flows— critical in extracting the maximum useful work out of the engine. Because of their high performance payoff and their relatively low development costs, seals have repeatedly shown high performance-to-cost benefit ratios in recent studies (Smith, 1994; Munson, 1994).

The objective of this paper is to provide an overview of the future engine development trends, provide an overview of the advanced seal technology requirements and highlight results of a detailed case study showing specific benefits of implementing advanced seals.

Advanced Subsonic Technology Program Goals

Operating Cost Reduction: Cost-conscious airline operators continue to demand lower operating costs including reduced fuel burn. Starting with a "clean sheet" aircraft/engine design, reduced engine weight translates into reduced overall system weight (engine and airframe weight). Reducing airplane weight and size reduces acquisition costs including interest. Costs to operate a 50 passenger regional aircraft are broken down as a percentage of total DOC in Fig. 1, showing that engine and airframe acquisition costs are 42% of the total costs. Engine maintenance costs amount to about half of the 22% total engine + airframe maintenance costs shown. Fuel burn amounts to about 19% of DOC, therefore reducing fuel burn results in additional savings over the life of the aircraft. Reduced fuel burn also helps lower emissions throughout the mission - key to a cleaner environment.

Specific fuel consumption (SFC) has continued to decrease over the course of turbine engine history, as shown in Fig. 2. In this chart a number of turbojet and turbofan engine SFC are shown versus engine certification date. Reducing SFC by 8% from current baseline engines will result in engines with SFC values for large engines to about 0.48.

Engine Pressure Ratio: For both engine classes, the goal is to increase pressure ratio roughly 50% from engines currently being certified for flight. For instance, the PW4084 now certified for the Boeing 777 has an overall pressure ratio of 38. Pressure ratios for large engines are targeted to increase to 50 or 60 to 1. Regional aircraft engines pressure ratios are expected to climb to 35 to 45. Engines decrease in size with increasing pressure ratios. Reducing engine size again reduces weight and acquisition costs.

As a result of the increasing pressure ratios, engine compressor discharge temperatures are expected to reach 1300 °F and higher, requiring compressor materials with 1450 °F operating temperatures (considering margin requirements). Turbine inlet temperatures are expected to be several hundred degrees over today's, requiring improved materials and thermal barrier coatings in the turbine hot section. Seal material temperatures will increase in proportion to the compressor and turbine material temperatures.

NASA/industry studies have indicated that improving engine performance and reducing operating costs would result in airline economies over the life of the engine fleet 20 times the total development cost (Smith, 1994)—a strong motivation for implementing advanced technologies that are cost effective.

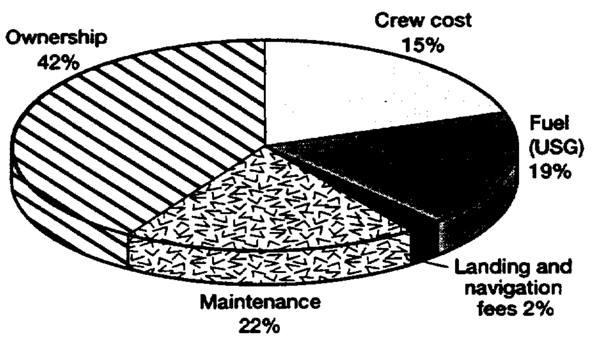


Figure 1.—Direct operating cost breakdown for a 50 passenger regional jet, 800 nautical mile mission.

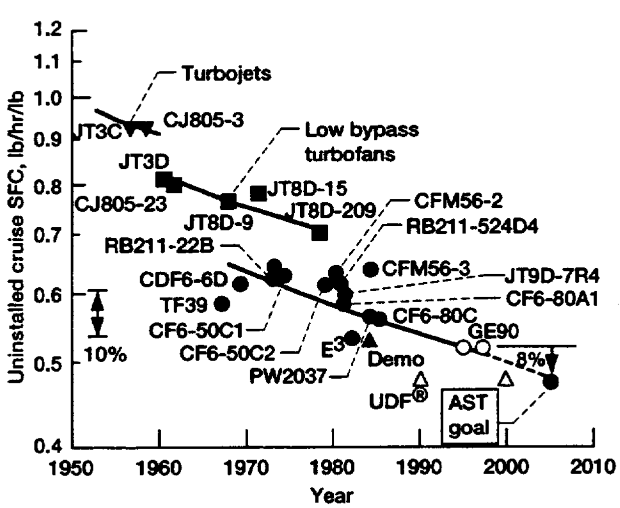
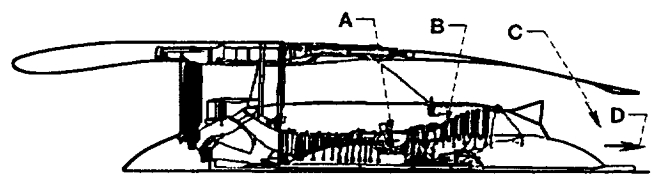


Figure 2.—Subsonic engine historical trend and program goal specific fuel consumption (Steinetz and Hendricks, 1994).

Motivation for Advanced Seal Development

Source of Engine Efficiency Gains: Overall engine efficiency, the useful work produced by the engine divided by the fuel energy, can be determined by the product of the three efficiencies illustrated in Fig. 3, (Smith, 1993). Plotting the historical trends of core efficiency versus the product of transmission and propulsive efficiencies, (Fig. 4), one sees that



Core thermal efficiency = $\dfrac{\text{Energy at the LP turbine entry}}{\text{Energy of fuel}}$ $\dfrac{(B)}{(A)}$

Transmission efficiency = $\dfrac{\text{Energy at the nozzle}}{\text{Energy at the LP turbine entry}}$ $\dfrac{(C)}{(B)}$

Propulsive efficiency = $\dfrac{\text{Useful work produced by engine}}{\text{Energy at the nozzle}}$ $\dfrac{(D)}{(C)}$

Overall efficiency = Core thermal x transmission x propulsive efficiency

Figure 3.—Breakdown of engine efficiencies (Smith, 1993).

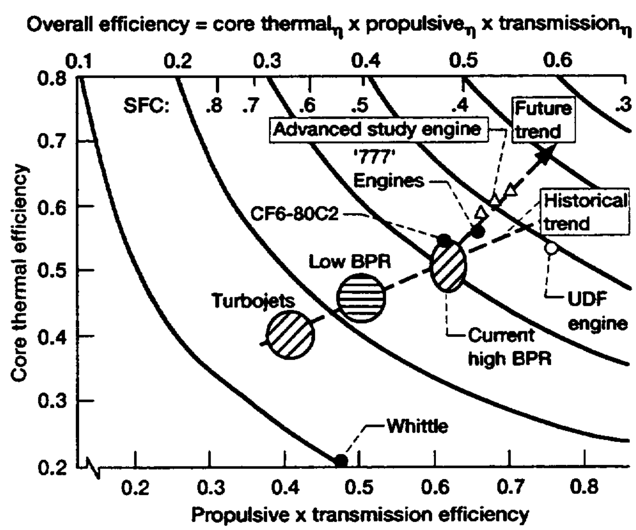


Figure 4.—Historical turbine engine overall efficiency as a function of core thermal efficiency and propulsive x transmission efficiency showing required improvements in core efficiency (Smith, 1993).

relatively more progress has been made recently in advancing the propulsive efficiency (e.g., fan) and low-pressure turbine than has been made in the core. This is illustrated by the "historical trend" line. To obtain the low SFC desired for the advanced engines, more progress must be made in increasing core efficiencies, as illustrated by the "future trend" line. As can be seen comparing the slopes of these two lines, reaching the SFC goal requires proportional, or balanced, increases in core and the product of propulsive and transmission efficiencies.

Increasing core efficiencies will be obtained by high cycle pressure ratios and compressor exit temperature, higher turbine inlet temperature, and improved component efficiencies including better sealing and secondary air flow management. Higher propulsive efficiencies will be obtained through increased bypass ratio and reduced installation drags and losses. Higher transmission efficiency will be achieved through improved fan system efficiencies and improved low-pressure turbine efficiencies.

Benefits: In addition to the strong case made for improving core efficiencies by reducing leakage and better managing secondary air flow systems, there are several other compelling reasons to advance seal technology to meet advanced engine goals. There is a strong correlation between the percentage reduction in seal leakage and either the percentage decrease in SFC or the percentage increase in thrust—all other things held constant. Advancements in seal technology generally are made with investments much smaller than those required for a compressor or turbine stage redesign and qualification. Studies performed by Stocker (1977) and corroborated by Smith (1994) estimated that making the same performance improvements with compressors or turbines would come at a cost a minimum of 4 to 5 times higher than the same improvements made by improving seal technology. Limited technology development budgets are compelling NASA and the engine community to exploit technologies such as seals with a high return on technology dollar invested.

Seal Technology: Current and Advanced Requirements

In examining seal requirements for advanced engines it is instructive to review current engine seal capabilities (Steinetz and Hendricks, 1997). Table 1 provides an overview of current engine seal capabilities in terms of pressures, speeds, temperatures, and materials. Table 2 summarizes expected seal operating requirements for next generation turbine engines including some military applications. Seals will generally be expected to operate hotter, seal higher pressures (to accommodate higher pressure ratios), and operate with higher surface speeds.

Carbon Face Seals: Face seals play a vital role in sealing bearing locations in turbine engines and auxiliary power units. Carbon face seals have low leakage and can seal pressures up to 150 psid. They currently operate reliably to speeds of 475 ft/s with acceptable friction and wear rates. These seals operate reliably with low leakage and cost less than labyrinth seals and therefore will continue to play a role in advanced aircraft engines. Even more will be asked of carbon face seals in future engines. Carbon face seal speeds will likely rise to 600 ft/s for advanced engines.

Research efforts are being directed at overcoming problems of face seal coking and blistering. AlliedSignal has had success replacing the carbon face seal with a ceramic ring seal overcoming the coking problems, eliminating oil odor in the cabin, and significantly increasing seal life (Ullah, 1997).

Labyrinth Seals: Perhaps the single most common flow path seal used over turbine engine history is the labyrinth seal. The labyrinth seal consists of multiple knife edges (typically 5) run in close clearance (0.01-0.02 in.), depending on location. Labyrinth seals can be configured in many ways including stepped and straight. Labyrinth seals are clearance type seals and therefore have high leakage rates. Furthermore, labyrinth seal leakage increases over time. Clearances open when shaft excursions force the labyrinth teeth into the adjoining rubstrips. Labyrinth seals are used as shaft seals, turbine rim seals, and as inner air seals—sealing the vane-to-drum inter-stage locations.

Table 1.—Current Turbine Engine Seal Technology

Seal	ΔP (psid)	Temp (F)	Surf. Speed (ft/s)	Materials
Face	150	1000	475	Carbon
Labyrinth	250-400	1300	1500	Ni Superalloy Teeth + Abradable
Brush	80-100/ stage	1300	1000	Cobalt Superalloy
Outer Air Seals: Compressor HP Turbine	Stage ΔP	1200 2000+	1200 1500	Abrasive Tipped Blades vs: Abradable Graded Ceramic

Table 2.—Advanced Turbine Engine Seal Technology

Seal	ΔP (psid)	Temp (F)	Surf. Speed (ft/s)	Materials
Face: Single Rot. Counter-Rot.	60 60	1000 1000	600 1000	Carbon Carbon
Film Riding Seal	800	1500	1200+	Ceramic + Superalloy
Labyrinth	250-400	1300	1650	Ni Superalloy Teeth w/Abrasive Tips + Abradable
Brush	140/ stage	1500	1500	Superalloy or Ceramic

Brush Seals: Significant efforts are underway to develop brush seals (Arora, Proctor, 1997; Short et al., 1996; Hendricks et al., 1994a; Chupp, Nelson, 1990; Holle, Krishnan, 1990). Brush seals consist of a dense pack of bristles sandwiched between a face plate and a backing plate (Ferguson, 1988). A primary attribute of the brush seal is its ability to accommodate transient shaft excursions and return to small running clearances, unlike labyrinth seals that wear to the full radial excursion. Brush bristles are oriented to the shaft at a lay angle (generally 45 to 55 degrees) that point in the direction of rotation. Brush seals are designed initially with a small radial interference ≤0.004 in. Starting with a small radial interference allows seal to accommodate seal-to-shaft centerline manufacturing variations providing a good seal. Leakage rates on initial run can be as little as 10-20% of comparable labyrinth seals. Seal temperatures are generally 1300 °F or less and surface speeds are generally 1000 ft/s or less.

Experience has shown that during engine operation, brush seal flow rates do increase due to wear. PW has entered revenue service with brush seals in three locations on the PW 4168 (see Fig. 5) for Airbus aircraft and on the PW4084 for the Boeing 777. PW has made leakage comparisons of new and aircraft- engine-tested brush seals and concluded that performance did not deteriorate significantly for periods approaching one engine overhaul cycle of 3000 hr. (Mahler and Boyes (1995)). Of the three brush seals examined, the "worst-case" brush seal's leakage doubled compared to a new brush seal. Even so, brush seal leakage was still less than half the leakage of the labyrinth seal.

Standard single stage brush seals typically are manufactured using 0.0028 in. diameter bristle wires. Bristle pack widths are usually maintained around 0.03 in. and the backplate is in contact with the last row of downstream bristles. Multiple brush seals are generally used where large pressure drops (≥80 psid) must be accommodated. The primary reason for using multiple seals is not to improve sealing but to reduce pressure-induced distortions in the brush pack, namely axial brush distortions under the backing ring, which cause wear. Also, researchers have noticed greater wear on the downstream brush if the flow jet coming from the

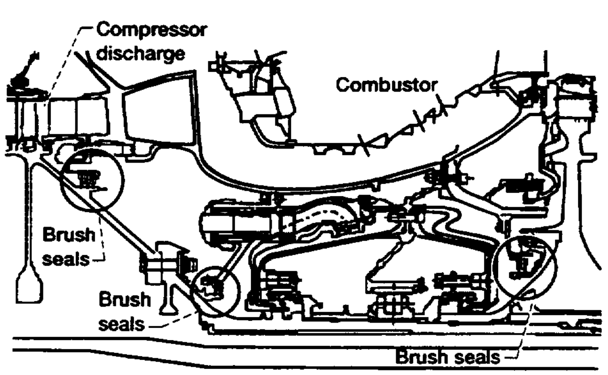


Figure 5.—Brush seals used in PW4168 engine. (Mahler and Boyes, 1995).

upstream brush is not deflected away from the downstream brush-rotor contact. Single stage brush seals made of larger diameter wires and thicker pack widths are showing promise of overcoming this effect.

Brush seal designs for higher pressure applications require bristle packs that have higher axial stiffness to prevent the bristles from blowing under the backing ring. Recently Short et al., (1996) amongst others have developed brush seals rated for pressure differentials of 120 psid and above using 0.006 in. diameter wire bristles, a thicker brush pack width (0.05 in.), and a front flexi-plate to limit bristle blow-down.

Brush seals will continue to evolve to meet the ever more demanding conditions required of them. Surface speeds will continue to increase in advanced engines. Surface speeds may climb as high as 1500+ ft/s in advanced engines, at temperatures up to 1500 °F. Under these extreme conditions, designs that would significantly limit the irrecoverable bristle wear are highly desirable. Other proprietary designs are also being investigated. Ceramic brush seals are also being investigated by a number of researchers including Hendricks et al., (1994b) and Howe (1994). Though not yet proven, hard ceramic bristles may be more wear resistant and may offer longer term wear lives.

Film-Riding Seals: Film riding seals rely on a thin film of air to separate the seal faces and show promise of reducing wear and leakage to its practical limit. Film riding face seals can be designed to operate at the high pressures and temperatures anticipated for next-generation gas turbine engines. There are two classes of film riding seals being developed for gas turbines: hydrostatic and hydrodynamic seals. Hydrostatic face seals port high pressure fluid to the sealing face to induce opening force and maintain controlled face separation (see Steinetz, 1998 review article). Changes in the design clearance results in an increase or decrease of the opening force in a stabilizing sense. Converging faces are used to provide seal stability. Hydrostatic seals are not applicable to lower pressure differential applications. Hydrostatic face seals suffer from contact during startup, requiring faces made of rub-tolerant materials.

The aspirating hydrostatic seal under development by GE and Stein Seal provides a unique fail-safe feature (Hwang et al., 1995, Wolfe et al., 1996, Bagepalli, 1996). The seal is designed to be open during initial rotation and after system shutdown—the two periods during which potentially damaging rubs are most common. Upon system pressurization, the aspirating teeth set-up an initial pressure drop across the seal that generates a closing force to overcome the retraction spring force causing the seal to close to its operating clearance (nominal 0.0025 in.). System pressure is ported to the face seal to prevent touch-down and provide good film stiffness during operation. At engine shutdown, the seal pressure across the seal drops and the springs retract the seal away from the rotor preventing contact.

Hydrodynamic or self-acting face seals incorporate lift pockets to generate a hydrodynamic film between the two faces to prevent seal contact. Munson (1992a,b, 1993) has developed a compressor discharge face seal for high-pressure, high speed applications and follows the early work of Ludwig (1978). A number of lift pocket configurations are employed including shrouded Rayleigh step, spiral groove, circular grooves and annular grooves (Steinetz, 1998). In these designs, hydrodynamic lift is independent of the seal pressure; it is proportional to the rotation speed and to the fluid viscosity. Therefore a minimum speed is required to develop sufficient lift force for face separation. Hydrodynamic seals operate on small ($\leq$0.0005 in. nominal) clearances resulting in very low leakage compared to labyrinth, or brush seals as shown in Fig. 6, (Munson, 1993). Because rubbing occurs during start-up and shut-down, seal faces are made of rub-tolerant materials.

Outer Air/Blade Tip Seals: Better management of blade tip leakage improves engine designs in several ways. Reduced compressor blade tip leakage improves compressor efficiency and improves stall/surge margins, improving engine operability. Maintaining tighter clearances over the life of the engine addresses a key observation that 80-90% of engine performance degradation is caused by blade tip clearance increase (O'Sullivan, 1994). In a limited number of commercial engines, blade tip clearance control is used. Blade tip clearance control is performed by preferentially cooling the turbine case during cruise operation. This has been successful in greatly reducing turbine blade clearances in the PW4000 series of engines and has resulted in handsome turbine efficiency gains (O'Sullivan, 1994).

Currently the industry does not use active feedback control. Adding feedback control by sensing average blade tip clearances and regulating case coolant will provide extra benefits, including allowing use of clearance control for other than cruise-only conditions, as is the case today. Mechanical control techniques are also being examined. Allison has demonstrated centrifugal compressor efficiency gains up to 1% using an experimental electromagnetic actuator to control compressor clearances (Weimer, 1992). For both active feedback control techniques, a pacing technical issue that is being worked is the development of reliable, high temperatures sensors.

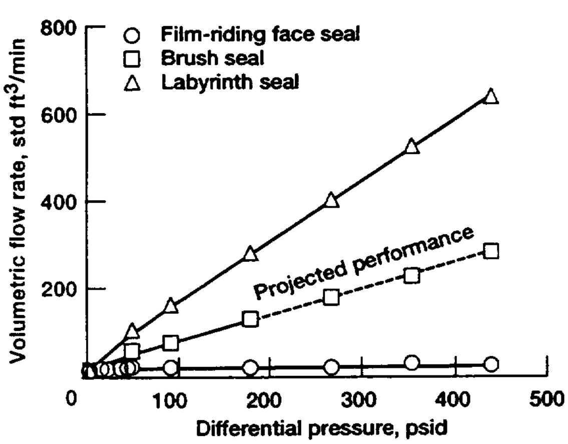


Figure 6.—Comparison of seal leakage rates as a function of differential pressure. Seal diameter 5.84 in. (148 mm). (Munson, 1993).

Loss of design clearances results in a loss of thrust, requiring an increase throttle setting to achieve the same engine performance. The increase throttle setting, however, increases the exhaust gas temperature (EGT) and thus reduces the life of the hot turbine components. When EGT exceeds a Federal Aviation Administration (FAA) certified limit, engine overhaul is required costing typically over $1 million (1998 dollars). Manufacturers will continue to develop techniques to combat this performance degradation to serve their cost-conscious airline customers.

Allison Engine 3007 Advanced Seal Study

The engine used for the advanced seal/secondary air management study was the Allison AE3007, 7000 pound thrust class turbofan engine, shown in Fig. 7. Although this is a modern engine in every respect, the secondary airflow system differs little from what is typical within the industry. In other words, very little has changed with regard to the management of the secondary airflow system in gas turbine engines. Over the same time period vast resources have been spent on materials development programs and investment in compressor, combustor, and turbine development programs. The reward has been steady improvement in overall engine performance.

More aggressive engine cycles place increasingly more severe demands on the secondary airflow systems and seals which will be used in these and future advanced engines. Even though the operating environment for these seals is becoming more severe, better performance from the overall secondary flow systems is needed so as not to erode performance gained by going to higher pressures and temperatures.

Study Goals: The study which is described in this paper serves to demonstrate that investment in sealing technology can provide substantial engine performance benefits. The goals of this program are to:

- Estimate SFC and other increased performance benefits possible as a result of incorporating advanced seal technology into advanced turbofan engines.
- Examine limitations in the current state of the art sealing technology for high compression ratio regional transport engines.

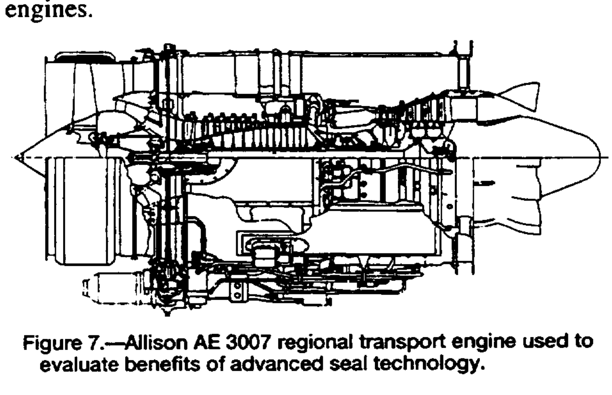


Figure 7.—Allison AE 3007 regional transport engine used to evaluate benefits of advanced seal technology.

After many years of little progress, there is renewed interest in development of improved gas turbine sealing systems. These efforts have successfully demonstrated advanced sealing concepts which are ready for further development and incorporation in advanced engine demonstration programs. These and other sealing technologies provide a relatively low cost way to greatly increase engine performance.

Approach

Several engine analysis groups contributed to this study to evaluate system level performance benefits and to investigate the feasibility of the approaches considered. Using the engine layout and cycles provided by the performance group, the flow systems group constructed an analytical model of the entire secondary airflow system. The model was then used to determine the airflow distribution within the engine. Turbine and compressor aerodynamic groups determined component efficiency versus seal clearances. The next step was to convert change in engine performance into change in overall aircraft performance. This is done by constructing a so-called "rubber" airframe model using Allison's mission analysis computer code. Mission studies were performed for a 50 passenger regional jet, powered by two AE3007 engines.

The relative sensitivity of an engine to the effects of leakage air is dependent on the particular cycle selected for the engine as well as the particular operating point selected. Results have been reported based on a hypothetical "mission," that was condensed from several different missions and operating points into a few typical power settings (See Table 3). For example idle is meant to represent ground idle, taxi, flight idle and descent. Maximum power represents both climb and takeoff. The next step was to define which secondary airflows constituted a leak. A consistent definition was required to enable comparison between baseline and advanced engines, particularly where there were differences in the secondary flow system. For this study, a leak was defined as any airflow which exits or enters from the main flow stream. Deliberate compressor bleeds were ignored, as well as all airflows which were associated with turbine blade or vane cooling. It was assumed that these cooling airflows had been sized exactly to perform their necessary functions. This results in a definition which allows engine comparisons to be made regardless of the configuration of the secondary flow system.

Table 3—Performance Results are Reported on the Basis of a Generic Regional Aircraft Mission

Operating Point	Condition	Duration (% of total)
Idle	SLS 103°F	34%
Cruise	20,000 ft, Mach = 0.7	33%
Max.	SLS (ISA+18°F)	33%

SLS = Sea Level Static
ISA = International Standard Atmosphere

Operational concerns enter into this decision as well. All present secondary flow systems bleed air from the compressor to provide pressurized air to the turbine interstage cavities. This air is used to prevent the ingress of hot flowpath gasses, to cool the disks and blades, and to balance thrust. Air is metered into the interstage cavities by the secondary seals which are located inboard, towards the disk hubs, and thus meter air up-stream of the blade/vane gap outboard at the disk rims. These vane/blade gaps then represent the real leak in the system.

Using the leakage definition, the engine was evaluated stage-by-stage to rank the relative sizes of the leaks. Thirty total locations were investigated. Table 4 provides the relative size of the various leakage flows, for those flows amounting to greater than 5% of the total weighted leakage. The mission ranking was obtained by dividing all the leakage flows by the highest leakage obtained for a particular operating point. These individual rankings were then multiplied by the percent of the mission spent at that particular point, then the weighted rankings were summed to obtain the overall mission ranking. The mission weighted percent represents the weighted fraction of total leakage that each individual leak represents. Compressor inner-band and turbine and compressor blade tip leakages effect component efficiency, but the leakage flows remain in the main flow stream. Design data was used to estimate engine blade tip and compressor inner band clearances throughout the operating range represented by the selected mission. These leakages did not fit the leakage definition as previously stated. They do impact performance and so were treated separately.

One notices from Table 4 that relatively few locations, primarily the turbine rim seal locations, are responsible for the majority of the total leakage. The top five locations are responsible for 75% of the total leakage. Table 4 also presents the "ideal SFC benefits" for each of the leaks. To attain these full benefits a "clean-sheet" design would be required. The preliminary design performed herein accepted as a ground-rule that there would be no changes allowed to the blade air flow circuit (e.g., no redesign of the blades). A "clean-sheet" or full hot-section redesign exploiting all of the benefits of advanced seals would have yielded even greater benefits than those indicated in the Benefits Section below.

Table 4.—Seal leakages accounting for more than 5% of total weighted leakage

Leakage Location	Mission Weighted Ranking	Weighted % of Total Leakage	Ideal SFC Benefit (%)
1V-1B Inner	0.96	25	1.25
2V-2B "	0.98	25	0.97
2B-3V "	0.39	10	0.46
3V-3B "	0.41	10	0.42
4V-4B "	0.24	6	0.19
2B-3V Outer	0.25	6	0.01
Total		82	3.30

Note: 1V-1B = first stage vane -first stage blade rim seal interface

Advanced Sealing Systems: Preliminary Design

Small Change Coefficients: Using the Allison mission analysis computer model and the engine cycle information small change coefficients were generated for all the engine bleed flows. These were used to determine the effect on overall aircraft/engine performance caused by changes in seal leakages. Component (compressor and turbine) small change coefficients were also generated to enable the effects of compressor inner band leakage, and blade tip leakage to be assessed. The compressor and turbine aerodynamic groups used analytical and test data from developmental engine testing to define the effect that these seals had on overall component efficiency. The small change coefficients are shown in Table 5. As an example how to use this table, a one percent change in compressor discharge to high pressure turbine airflow results in a 0.7% change in SFC.

Based on the results of the previous steps it was possible to rank the various leakage locations to determine where the largest performance gains could be obtained for several approaches considered.

Approaches Considered: Several proposed designs were analyzed using a proprietary compressible flow computer code to analyze secondary air flows. Four turbine rim seals were planned initially (Fig. 8, iteration A). These were placed into the secondary flow model with no other adjustments made to the flow system relative to the existing AE3007. This iteration resulted in a net leakage reduction of 0.75% of core flow. Representative film-riding seal flow characteristics derived from experimental and analytical data were provided to the secondary flow code. It was clear more gain could be realized. The cavity pressures increased due to more effective rim seals. This caused large air leakage in places which previously had ingress from the gas path. In addition, a large amount of high pressure air was leaking into the low pressure cooling circuit. Adjustments were made to reduce airflow into the HP turbine cooling circuit, as well as changing from segmented to solid turbine blade cover plates.

A second major iteration (B) removed the second-blade-third-vane (2B-3V) rim seal and added two film riding face seals. This was an effort to prevent leakage of the HP air into the LP cooling circuit. This produced an overall leakage reduction of 2.75% and did totally separate the HP and LP cooling circuits. Unfortunately the amount of air required through the compressor ID rotor bleed to feed the LP circuit was excessive. A third iteration (C) was considered to solve the compressor problem. Although this probably would have resulted in even greater leakage reduction the proposal was not evaluated due to flight safety concerns, and the sheer complexity and risk trying to introduce so many advanced seals at one time into the engine.

Mechanical Configuration Selected: The fourth configuration (D) was selected that uses 3 advanced film riding face seals to seal the 1V-1B, 1B-2V, and the 2V-2B locations. This configuration balances complexity and risk against the benefits of reduced leakage and increased performance. Sealing just the 3 locations saves 2% in total leakage flow (1.2% chargeable; 0.8% recoverable). Using existing proprietary design methodology Allison developed space claims for the proposed seals which were located as close to the main flow path as possible. The resulting mechanical arrangement is shown in Fig. 9. Notice that there are still very small cavities outboard of the seals. These must be purged with some cooling flow. While no cavity is the most desirable situation, this is the best that could realistically be accomplished.

For this preliminary design exercise, candidate film riding face seals were designed based on the expected operational requirements which include: operating temperature, pressure,

Table 5.—Small change coefficients used to evaluate the effect of leakage on overall regional aircraft performance

	ΔSFC %/PT	ΔFn/Wgt %/PT	ΔTOGW %/PT	ΔMFB %/PT
CDP -> o/b	+1.5	-2.8	+0.4	+2.0
CDP -> HPT	+0.7	-2.1	+0.2	+1.1
INT -> HPT	+0.3	-1.6	+0.2	+0.6
η HPC	+1.0	-1.9	+0.2	+1.3
η HPT	+0.8	-0.9	+0.1	+1.0
η LPT	+0.8	-0.9	+0.1	+1.0

CDP = Compressor discharge
o/b= overboard
η = efficiency
HPC = High pressure compressor
LPT = Low pressure turbine
HPT = High pressure turbine
SFC = Specific fuel consumption
PT = percentage point
Fn/Wgt = Thrust/weight
TOGW = Take-off gross weight
MFB = Missionized fuel burn
INT = Compressor Interstage (10^{th})

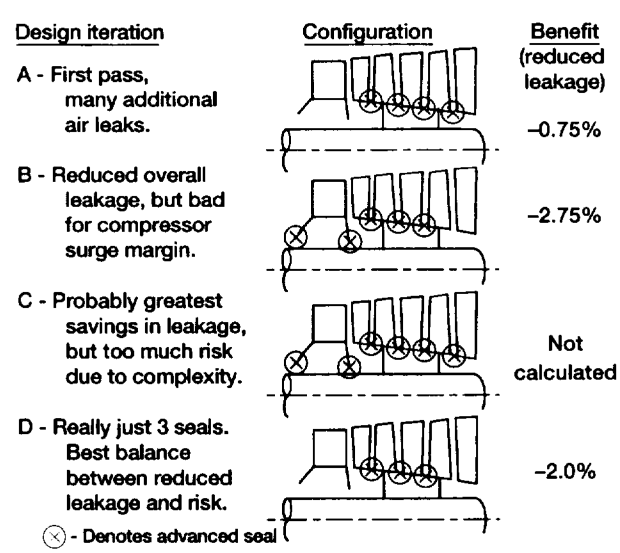


Figure 8.—Several different advanced sealing configurations were studied before selecting one for further analysis.

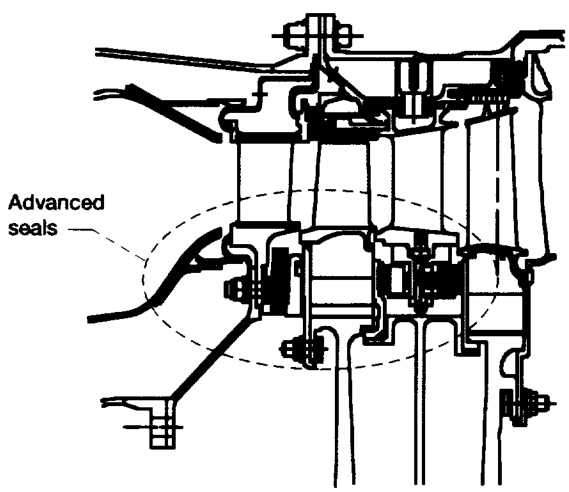


Figure 9.—Turbine rim seal mechanical configuration.

speed, relative thermal growths of rotor and stator, part tolerances, and anticipated thermal and mechanical deflections. All of these parameters effect the seal design and in particular the required space envelope. The size of the seal determines to a large extent how close to the flow path the seal can be physically be located. Allison and NASA seal codes (Shapiro and Athavale, 1994) were used to develop preliminary designs for the candidate advanced seals used for these studies.

Every attempt was made to ensure that the seal substitution was being performed in a realistic manner. Heat transfer work was done to verify that turbine metal temperatures remained within presently accepted maximum operating temperatures. The effects of advanced seals were evaluated by substituting flow characteristics for the advanced seals in selected sealing locations. In most cases this required some additional rebalancing of the secondary flow systems because of the large reductions in leakage. During this process it was evident that as a result of improved sealing of the secondary flow systems, the engine could be redesigned to reduce fuel burn or increase thrust; or by reducing rotor inlet temperature, increase engine life.

Analytical Studies and Discussion

After selecting the mechanical configuration, a number of more detailed analytical studies were performed to assess the feasibility of the approach.

Heat Transfer Analysis: The purpose of these studies was threefold: fine tune the air flows, make sure that predicted metal temperatures were within acceptable limits, and prepare temperature files for subsequent use by stress analysis. Temperatures were calculated initially for a steady state power point, and then later for transient operating conditions.

Several studies were used to determine if the small cavities above the seals had to be completely purged or if some ingress could be tolerated. When leakage was adjusted to completely eliminate ingress, most of our gains disappeared. This produced metal temperatures below current AE3007 values and the cavities were being overcooled. Next we tried leakage flows halfway between the first trial and the low value possible for film riding seals. These results also indicated that we were closer to the AE3007, but still overcooling in some locations. Finally we tried the film riding seal leakage values. The team agreed that these results were acceptable from an operational standpoint with one exception, the 2V-2B cavity. This cavity was running slightly hot. The reason was due to leakage through a segmented blade cover on the aft side of the second rotor. This was changed to solid, and the problem was eliminated.

Results of the thermal analysis for the maximum power point are shown in Fig. 10. This figure plots temperature *differences* of the advanced engine relative to the current AE3007. There are a umber of interesting results presented which bear pointing out. There was less than a 10 °F change in turbine wheel bore and web temperatures indicating that the new seals have almost no effect on the bulk of the rotor.

Figure 10 shows that the aft first vane platform and the first blade platform forward are considerably hotter (e.g., 270 °F shown) than on the existing AE3007. However, the actual peak first vane metal temperature is only 35 °F hotter than the existing AE3007. This temperature rise is well within the present capability of the existing vane. Peak metal temperatures for the first blade are almost identical between the existing AE3007 and the new engine. What Fig. 10 is showing is essentially a redistribution of temperatures in the first blade.

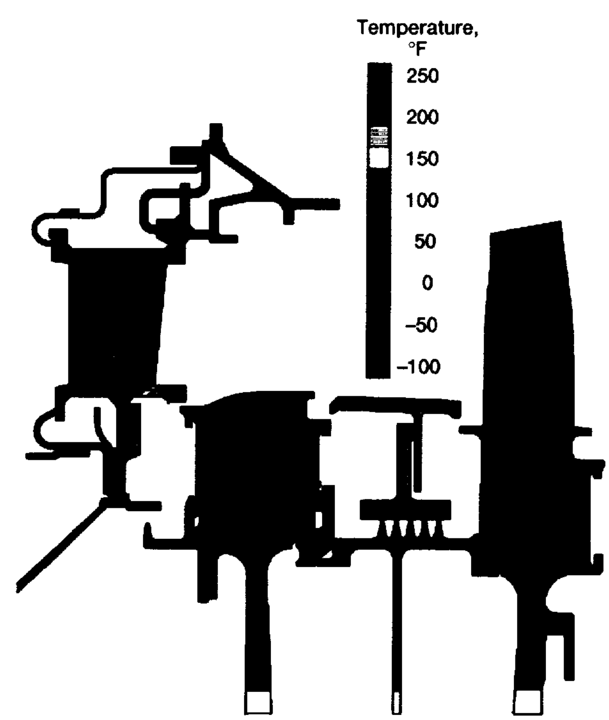


Figure 10.—Advanced HP Turbine temperatures with film-riding face seals minus AE 3007 temperatures. Conditions: steady-state Max T/O; 73 °F; 175 KCAS.

The reason for these temperature differences can be traced both to an overall reduction in airflow, and changes in the leakage pathways brought about by the increased effectiveness of the new 1V-1B seal. The new seal is much more effective than the existing labyrinth seal at restricting flow through the 1V-1B gap. By providing such an effective seal at this location we reduce the amount of air which can flow through this gap. Temperatures thus rise outboard of the seal as expected, accounting for the higher blade and vane platform temperature in the vicinity of the small cavity outboard of the first advanced seal.

Despite the fact that the differential pressure across this seal is small, airflow through the current labyrinth is substantial. In effect this leakage flow presently provides more cooling than is needed at this location. We also found that we needed to reduce airflow to this inter stage cavity by approximately 0.8%, and to provide solid blade cover plates. If this was not done the air had a tendency to flow through the blade attachment regions outward in the main flow path, and axially into the 1B-2V cavity. Figure 10 shows that the 1B attachment is much cooler with the new system than the existing AE3007. This is the effect of the remaining residual airflow through the first blade attachments. Thus, temperatures of the first vane platform aft, first blade platform forward, and first blade attachments are linked to both the magnitude and the direction of the leakage flows from this inter-stage cavity.

Failed Seal Scenario: The next step in the thermal analysis was to look at the effect of failed seals. This study was also done at maximum power, having the highest boundary temperatures. We had considered the possibility of single or multiple seal failures when conducting the secondary flow studies. A design guideline followed was that any failed seal scenario would not effect the flow of cooling air to the blades and vanes. As stated previously however, the fact that the total air flow to the seals had been reduced meant the possibility of ingress existed at these locations in the event of a failed seal. We simulated failed seals by substituting leakage flows of conventional labyrinth seals in place of the advanced seal flows. The labyrinth seal gap modeled is consistent with what the AE3007 presently runs with existing hardware. This could easily be provided by a back-up seal which would control flow to these similar levels.

Failure of the first stage seal had an almost negligible effect since the differential pressure at this location is only slightly biased in favor of flow out of the inter-stage cavity into the flow path. Failure of the 1B-2V-2B seal results in considerable ingress to the 1-2 cavity outboard of the spacer. However no change in blade cooling flow was observed. Rotor and blade attachment temperatures showed only a slight rise in temperature due to the mitigating effects of the blade cooling flows. A 300 °F rise in temperature of the second stage vane inner band and turbine 1-2 spacer was observed. While this is not an immediate concern, it would require engine removal and overhaul at some point. Failure would be detectable by a drop in power and a rise in maximum measured gas temperature. By adding monitoring thermocouples on the second stage vane, an additional indication of failure would also be provided. Gas temperature is currently measured near this location, so this approach is feasible.

Structural Analysis: A thermal-structural finite-element analysis was performed on the HP rotor systems. At this preliminary design stage, some of the parts did not quite meet Allison's life criterion. However in the judgment of the design and analytical groups involved the results are close enough that the parts could be easily modified during a detailed design. Design changes would involve material substitution, adjusted fillet radii, and other similar minor details.

Seal Tracking: Using finite element analysis coupled with engine limit stack and tolerance data we were able to predict relative axial, radial, and angular deflections between the rotor and stator. These are presented in Table 6. The deflections must be held relatively small so that the proposed seal designs are able to track. For long life and high reliability the seals must be able to operate in a non-contacting manner, with the possible exception of startup and shutdown.

Proprietary hydrodynamic seals are being developed at Allison and within the seal community that show promise of meeting the demanding seal operating conditions. A key feature of all of these seals will be seal compliance, namely the ability to accommodate and dynamically track thermal coning and circumferential out-of-flatness of adjacent seal surfaces. Depending on how this idea is implemented, a slight increase in leakage could be possible.

Reliability: Greater than 99% reliability as demonstrated through test programs is required before these seals would be considered for use on an engine. Component reliability problems are direct drivers of unscheduled maintenance. Unscheduled maintenance is a large driver of direct operating cost. Seals that are ultimately developed for this application will be designed based on the need for extreme reliability.

Another key goal is flight safety. No matter how reliable the individual components that comprise the engine, the total

Table 6.—Anticipated run-out and deflections, operational requirements at the proposed advanced rim seal locations

Location	1V-1B	1B-2V	2V-2B
Radial (in)	.04 to .09	.08 to .13	.02 to .10
Axial (in)	-0.04 to +.02	-.03 to +.13	-.04 to +.19
Angular (ID to OD) deg.	.13	.32	.52
Circumferential (TIR) (in)	.004	.006	.008
Max. Speed (ft/sec)	1150	1050	1050
ΔP (max. psid)	0.2	8.5	60
ΔP (min. psid)	0	0.5	6.3
Temp. (max. metal) °F	1200 - 1500	1100 - 1400	1100 - 1400

Note: 1V-1B = first stage vane - first stage blade rim seal interface system needs to be designed such that the failure of any one component does not result in catastrophic failures of any other part of, or the total engine. It is also important that any failure also be immediately detectable. Based on the nature of the failure the pilot can then make the determination to continue running till the plane can be landed safely, or shut down the engine so that secondary failures do not occur.

Benefits

As a result of installing the advanced seals in the HP turbine, secondary airflows were reduced by 2%. Of this 1.2% was chargeable providing for significant performance benefits. This savings in leakage can be used in several ways.

Increased Engine Power: For the same turbine inlet temperature and fuel burn, the reduced leakage can be used to provide more power from the same engine. Using the small change coefficients in Table 5 (CDP->HPT row), one can attain 2.5% (e.g., 2.1 x 1.2%=2.5%) higher engine thrust-to-weight ratio or 0.24% lower take-off gross weight.

Increased Operating Margin: In this case the maximum power output remains fixed. Lower leakage allows this power to be obtained using less fuel, and at a lower maximum cycle temperature. The sensitivity coefficients presented in Table 5 do not apply in this case—a new turbine map is developed and implemented in the performance computer code. Then the performance deck was run to match thrust levels between the original and the modified engines. These runs made it possible to compare fuel flow rates, specific fuel consumption, rotor inlet temperature, etc. and were directly attributable to the reduction in leakage.

The net result was a 0.9% reduction in SFC, and an approximately 20 °F average reduction in turbine inlet temperature across the range of operating points from idle to maximum power. This result was input into a detailed turbine blade life model which was itself based on a very detailed regional mission. The result of this was a greater than 50% increase in both first and second stage blade life. This in turn would result in a 6.6% decrease in engine maintenance costs associated with flying the 50 passenger regional aircraft mission.

Since engine maintenance costs are 11% of total operating costs, Fig. 1, this reduces direct operating costs by approximately 0.73%. Adding the contribution of SFC reduction (0.9% times 19% Fuel component, Fig. 1), brings the total reduction in direct operating costs to 0.89%, or 18% of the NASA AST goal, with just 3 seals.

Emissions: Reducing fuel burn and making the other cycle changes had a generally positive effect on emissions, using the ICAO standards for emissions calculations. Due to longer combustor residence time and higher compressor discharge temperature, a very slight 0.7%, increase in oxides of nitrogen (NO_x) was observed. Starting with a "clean sheet" design NO_x emissions would have been reduced. Carbon monoxide (CO) emissions were reduced 3.47% and unburned hydrocarbons were reduced 2.91%.

Summary and Conclusions

Engine designers are re-evaluating all aspects of turbine engines to meet the efficiency, performance and operating cost goals set for next generation turbine engines. A comprehensive survey was made of cycle losses in terms of leakages of a modern regional jet engine, the Allison Engine AE3007. The survey identified the largest performance gains come from applying advanced seals within the gasifier section of the engine. This is because energetic compressor discharge air must be used to purge the majority of the gasifier turbine. The study demonstrated applying advanced seals will lead to large reductions in specific fuel consumption and ultimately direct operating costs. Based on the investigations performed the following points are clear.

1. Incorporating just three advanced sealing devices, (e.g., mechanical face seals), in a gas turbine engine should improve engine SFC by approximately 0.9%. Use of additional advanced seals would provide even greater benefit with somewhat greater complexity.
2. Advanced sealing devices pay very large dividends. Just three advanced face seals installed at appropriate locations in the current AE3007 should reduce direct operating costs of the regional aircraft by approximately 0.89% or 18% of NASA's stated goal.
3. Analytically it was shown that the advanced seals could be incorporated into the gasifier section of a gas turbine engine. The resulting design required no compromise to the main flow path and the design meets all criterion for flight safety. During a detailed design, minor design modifications are possible to allow the hardware to meet all Allison design criterion for hot section life.

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13. ABSTRACT *(Maximum 200 words)*

Cycle studies have shown the benefits of increasing engine pressure ratios and cycle temperatures to decrease engine weight and improve performance in next generation turbine engines. Advanced seals have been identified as critical in meeting engine goals for specific fuel consumption, thrust-to-weight, emissions, durability and operating costs. NASA and the industry are identifying and developing engine and sealing technologies that will result in dramatic improvements and address the goals for engines entering service in the 2005-2007 time frame. This paper provides an overview of advanced seal technology requirements and highlights the results of a preliminary design effort to implement advanced seals into a regional aircraft turbine engine. This study examines in great detail the benefits of applying advanced seals in the high pressure turbine region of the engine. Low leakage film-riding seals can cut in half the estimated 4% cycle air currently used to purge the high pressure turbine cavities. These savings can be applied in one of several ways. Holding rotor inlet temperature (RIT) constant the engine specific fuel consumption can be reduced 0.9%, or thrust could be increased 2.5%, or mission fuel burn could be reduced 1.3%. Alternatively, RIT could be lowered 20 °F resulting in a 50% increase in turbine blade life reducing overall regional aircraft maintenance and fuel burn direct operating costs by nearly 1%. Thermal, structural, secondary-air systems, safety (seal failure and effect), and emissions analyses have shown the proposed design is feasible.

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